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AN ANALYSIS OF THE PERFORMANCE AND SPACE STORAGE CHARACTERISTICS OF SMALL PROPANE-FLOX UPPER STAGES

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SUMMARY

The use of hydrogen in rocket stages poses a significant insulation problem for missions involving space storage times greater than a few hours. There is a great lack of information regarding both the distribution of incoming heat in a hydrogen tank in zero g and the weight penalties associated with zero g propellant venting devices for long coast missions. This lack of information has increased interest in less energetic, but more space storable, propellant combinations.

The purpose of this study was to examine one such combination, propane-flox, for use in a multi-purpose upper stage in order to: (1) determine its space storage characteristics and payload capability, and (2) compare its performance capability with that of other propellant combinations.

The results of the study indicate that it is possible to design a propane-flox stage having a long space storage (no vent) capability. The performance of such a stage, however, is inferior to hydrogen fueled stages for high energy, short coast missions. For long coast, low energy missions, specifically the Mars orbiter mission investigated herein, the propane-flox stage displayed only a small performance advantage over an earth storable stage.

INTRODUCTION

Up to the present time, most space missions have been accomplished utilizing launch vehicles incorporating upper stages that employ either solid or "earth storable" propellants. However, as the demands on payload size and/or mission energy increase, it is appropriate to consider more energetic upper stages. This is evident in the current development of the Centaur, S-IVB, and S-II stages, all of which are designed to use the cryogenic propellant combination hydrogen-oxygen.

The use of hydrogen in rocket stages poses a substantial insulation problem for missions involving propellant space storage times greater than a few hours. For short duration missions, the heat transfer through the insulation and tank supports can be absorbed by bulk heating of the hydrogen. However, as mission times increase, a point is reached where bulk heating must be terminated (either the hydrogen density decrease has absorbed all of the tank ullage volume or the vapor pressure of the hydrogen reaches the maximum allowable tank pressure) and tank venting, either continuous or periodic, must begin.

Once venting begins, the primary concern is insuring that gaseous, not liquid, propellant is vented overboard. This imposes the requirement for a positive propellant control system that maintains the ullage bubble at the tank vent location. Both passive and active propellant control systems have been proposed for use during zero g flight. At the present time, insufficient experience has been obtained with any non-propulsive propellant positioning device with cryogenics to accurately determine its effectiveness and its associated vehicle system weight penalty.

Future experience may prove some of these devices to be effective and lightweight. If, however, the weight penalties associated with practical systems are substantial, then for some missions the use of a less energetic propellant combination, which does not require venting, may provide equal or better payload capability or greater mission reliability. For these reasons, an analysis was conducted to determine the stage characteristics and payload capability of an unvented upper stage utilizing the propellant combination propane-flox.

For the past several years, the Lewis Research Center has been investigating high energy upper stages for unmanned scientific missions. The purpose of that work has been to determine the feasibility of developing one, multi-purpose, high energy stage that could serve a variety of missions rather than developing a variety of upper stages to serve specific missions. Some of this work has been reported in reference 1 (TMX-52127). Based on these studies, a 7000-pound hydrogen-oxygen stage using an RL-10 engine (with the capability of substituting fluorine at a later date) has been identified as a desirable upper stage for both long coast and short coast missions.

Currently, most of the missions envisioned for the high energy stage do not have a long coast requirement, so tank venting will not be a problem. With this in mind, and adhering to the philosophy that a new upper stage should be a multi-purpose vehicle, the propane-flox stages were evaluated for: (1) a high energy short coast mission (solar probe) and (2) a long coast mission (220-day planetary orbiter mission where a deep-cryogenic stage would have to be vented). The analysis emphasized propellant management problems to determine if venting could be avoided through the use of propane-flox. NASA, for some time, has been investigating experimentally the performance of various light hydrocarbon fuels with flox mixtures. From this work, it appears that methane is the best choice for use in a completely transpiration cooled engine whereas propane or butene 1 display advantages in an engine using combined regenerative and transpiration cooling.

Propane, which is denser and has a wider liquid temperature range than methane, was selected for this study. Particular consideration was given to stage configuration, engine thrust, pressurization and propellant insulation requirements. Having optimized the characteristics of the propane-flox stage, its payload capabilities for the above two missions were determined and compared to those of deep cryogenic stages and an earth storable stage using the MMH-N_2O_4 combination.

STAGE DESIGN

Configurations, Structures and Materials

A structural design study of the propane-flox stage was undertaken in order to estimate stage structural weights for performance calculations and to select a feasible configuration based on stage inert weight, propellant heating, stage length and structural integrity. To minimize stage inert weight, a stage configuration of low structural weight was the first design goal. The second design goal was to minimize the heat flux conducted through the tank supports to the propellants to an acceptable level so that tank venting would not be required. The design of a compact stage of low overall height was the third design goal. A compact design helps to assure minimum structural weight and the low overall stage length minimizes launch vehicle integration problems since the addition of a long upper body to an existing launch vehicle could aggravate the flight loads or result in reduced launch availability. The fourth goal, structural integrity, is a qualitative judgment that was applied to the evaluation of all proposed configurations. The primary factors considered were thrust load paths and tank support.

All of the proposed configurations were constrained to a 120-inch maximum outside diameter. Inspection of potential missions indicated that a kick stage would usually fly off the Centaur vehicle so a 120-inch diameter stage would be desirable for transmitting loads to the Centaur forward support ring (station 219 ring). The Centaur and propane-flox stages are considered as upper stages for the Atlas and Saturn IB boost vehicles. The loads from a 120-inch diameter stage would be transferred to the Centaur by primarily axial components. Use of a larger diameter could induce radial loads on the Centaur, and this is not desirable for the pressurized, thin-shell Centaur structure. Also, a stage diameter of more than 120 inches would require a bulbous upper body and result in increased flight loads for flight on the Atlas-Centaur.

Consideration of the design goals and constraints led to two basic stage configurations. For each stage configuration, both pump-fed and pressure-fed engine configurations were studied. These configurations are described in the following paragraphs.

Four Sphere Configuration

The proposed four sphere configuration for a 6000-pound propellant load and a 5000-pound thrust pump-fed engine is shown in figure 1.

The propellant system consists of two spherical fuel tanks and two spherical oxidant tanks clustered around the engine. The storage of each propellant in two tanks would require manifolding of the propellant feed lines. The tanks would be insulated with multilayer insulation and supported from the structure by cylindrical fiberglass supports to reduce heat flux into the propellants. A high strength weldable aluminum alloy, 2219-T81, was selected for the tank material.

The configuration is a compact design of only 85 inches overall length. Thus, the length of existing launch vehicles would not be greatly increased by the addition of this kick stage. The stage total weight is estimated at 7400 pounds and the mass fraction is 0.81. A complete stage weight summary is shown in Table I. The weight summary presented in Table I (as well as in succeeding tables) includes weight estimates for guidance, electrical, attitude control, telemetry, tracking and range safety systems. These system weights were based on those reported in reference 1 with some modifications to reflect recent design effort.

A significant structural design feature of the stage is the cruciform beam which forms the thrust structure and provides principal support of the tanks. The beam is a built-up section of aircraft type construction. The cruciform beam also supports the engine actuators, helium pressurization bottle, and the cold nitrogen gas attitude control bottles. Each propellant tank is also supported by the 120-inch outside diameter ring that connects the quadrants formed by the beam. Thus, the beam and ring form a "spoked wheel" that supports all the stage components and transmits the engine thrust to the payload adapter. The structural material, like the fuel tanks, is 2219-T81 aluminum alloy (the use of beryllium-nickel was assumed for the flox tanks). The total structural weight is estimated at 250 pounds. A detailed structural weight summary is given in Table II.

Stage weight and mass fraction estimates were also made at 3000 and 9000 pound propellant loads for the pump-fed engine case. The design process was then repeated for the pressure-fed engine case at the same 3, 6, and 9K propellant loads. The results of these weight estimates are presented as jettison weight curves in figures 2 and 3. The jettison weight curves were then used to study performance and optimum propellant loading.

Comparison of the four sphere configuration with the design goals shows the stage length goal has been met. The stage is very compact and the tanks are not used as structural members. The design goals of low stage inert weight and zero propellant boiloff have not been met in a completely satisfactory manner. The mass fraction of 0.81 for the 6K pump-fed case indicates considerable structural weight. Thermal analysis shows the configuration would allow excessive heat flux into the flox for long duration missions. For example, on a Mars orbiter mission, the flox boiloff with one inch of multilayer insulation on the flox tanks would be 50 pounds. Increasing the insulation thickness to three inches would only reduce the boiloff by 4 to 46 pounds. This boiloff of propellant and subsequent tank venting are not consistent with the design goals. In summary, it is concluded that the proposed four sphere configuration is not an attractive configuration for the propane-flox propellant combination.

Two Oblate Spheroid Configuration

The proposed two oblate spheroid configuration for a 6000-pound propellant load and a 5000-pound thrust pump-fed engine is shown in figure 4. This configuration was developed primarily to solve the propellant boiloff problem experienced with the four sphere configuration. The LPG would be contained in the forward tank thus placing the relatively warmer propane (300°R) between the room temperature payload and the colder (156°R) flox stored in the aft tank. Both tanks would be covered with insulation and be supported from the structure by fiberglass cylinders to reduce heat flux into the cryogenic propellants. Thermal analysis indicates that propellant venting would not be required for long duration missions. This meets one of the design goals.

An attractive feature of this configuration is the simple propellant feed lines. A single line from each tank would supply the engine and no manifolding would be required.

The configuration is not a compact design as the overall length is 142 inches; approximately 57 inches longer than the four sphere configuration. The length of an existing vehicle would be significantly increased by the addition of this stage and the vehicle could require strengthening to maintain present launch availability. The stage total weight is estimated at 7315 pounds and the mass fraction is 0.820. The stage is 85 pounds lighter than the four sphere configuration. Less insulation and structural weight for the two oblate spheroids configuration accounts for the difference. A complete stage weight summary is shown in Table III.

A structural design feature of this stage is the utilization of the oxidant tank for transmitting the engine thrust load to the stage truss structure. The tank ullage pressure would be maintained sufficiently large to assure tensile membrane stresses and tank stability. A pressure-stabilized structure is an efficient technique for transmitting thrust loads, and this accounts for the structural weight advantage of this stage compared to the four sphere configuration. In addition, the propellant tanks of the four sphere configuration imposed a weight penalty on that stage. Tank thicknesses of eight mils were adequate from design considerations; however, a minimum gage of 15 mils was stipulated due to handling and manufacturing considerations.

The space truss structure consists of a forward and aft ring laced together with strut members inclined to the vertical. These inclined struts can transmit axial and lateral loads. The truss structure supports the LPG tank, payload, helium pressurization bottles, and other equipment. A high strength weldable aluminum alloy, 2219-T81, was selected for the truss structure and the propane tank while beryllium-nickel was selected for the flox tank and thrust cone. The total structural weight is estimated at 229 pounds and a detailed structural weight summary is given in Table IV.

Stage weight and mass fraction estimates were also made at 3000 and 9000 pound propellant loads for the pump-fed engine case. The design process was then repeated for the pressure-fed engine case at the same 3, 6, and 9K propellant loads. The results of these weight estimates are presented as jettison weight curves in figures 5 and 6. The jettison weight curves were then used to study performance and optimum propellant loading.

Comparison of the two oblate spheroid configuration with the design goals shows the stage length and structural integrity goals have not been met in a completely satisfactory manner. The stacked tank arrangement makes for a lengthy stage and the use of a pressure-stabilized structure is a more complicated technique than the cruciform beam of the four sphere configuration. The design goals of low stage inert weight and zero propellant boiloff have been met by the two oblate spheroid configuration. The mass fraction is somewhat better than the four sphere configuration and a thermal analysis shows that tank venting would not be required for long duration missions. Thus, each configuration meets some of the design goals. Since the two oblate spheroid configuration is feasible for long duration missions, it is concluded that this configuration is the more satisfactory configuration of the two proposed.

Thermal Control

The primary uncertainty in determining the non-vented space storability of a given propellant combination is the distribution of the heat entering the propellant tanks. If bulk heating of the liquid occurs, then the allowable space storage time can be (depending upon the magnitude of the heat flux and the maximum allowable tank pressure- several orders of magnitude longer than if a substantial portion of the heat entering the tank goes into vaporizing liquid. If, instead of depending upon bulk heating of the liquid, the stage can be designed so that the net heat transfer is out of the propellant (sum of radiation plus conduction from the tank greater than sum of radiation plus conduction to the tank) then, the stage will be space storable essentially independent of time. This was the approach taken in this study; that is, to arrive at a combination of stage configuration, insulation, and propellant temperatures such that the heat out of the tanks was equal to or greater than the heat into the tanks rather than to depend upon the heat capacity available in warming the propellants from some initial to final temperature to provide the storage capability. It should be noted that having adopted this approach, provision must be made for extremely long coast missions to insure that propellant freezing does not occur.

When selecting a thermal protection system for long duration missions requiring the storage of cryogenic propellants, the choice of multifoil insulation or a combination of multifoil and another insulation (e.g. foam) is inevitable because of the superior performance of multifoil over other insulations.

However, when it is desirable to eliminate the net heat transfer to the propellant tanks completely to avoid the necessity of venting, the following procedures, in addition to using multilayer insulation, are required:

1. Jettisoning the shroud surrounding the propellant tanks so the propellants can radiate to the cold environment of space.

2. Orienting the vehicle so that the propellant tanks are shadowed from the sun (e.g., payload toward sun).

3. Separating the warm and cold propellant tanks as far apart as possible and also placing the payload as far from the propellant tanks as possible in order to obtain less radiation heat transfer to the propellants.

4. Selecting a low-conducting support system to substantially reduce the heat transfer by conduction (e.g., fiberglass instead of titanium).

The assumptions used in the thermal analysis of the propane-flox kick stages were: (1) Tanks were insulated with multilayer insulation having a density of 5 pounds per cubic foot. To allow for degradation around tank supports and penetrations, the thermal conductivity values reported by Linde in reference 2 were degraded by a factor of two. (2) Payload was oriented toward the sun. (3) Tanks were supported in an open truss to take advantage of radiation to the cold environment of space. (4) Flox storage temperature was 156°R. (5) Propane storage temperature was varied from 250 to 400°R. (6) Payload was represented as a 10-foot diameter disc at 520°R.

It should be noted here that foam insulation was also used on the two oblate spheroid configurations as will be discussed later.

The modes of heat transfer considered were: (1) radiation interchange between the payload and propellant tanks, (2) radiation interchange between propellant tanks, and (3) conduction heat transfer through the supports into the propellant tanks.

Four Sphere Configuration

In calculating the radiation interchange from the payload, it was assumed one flox sphere was centered directly underneath the payload disc. (In the thermal analysis, one tank is considered at all times.) Then, by utilizing IBM 7094 computer programs, the radiation configuration factor between the payload disc and the flox tank and a radiation heat balance were calculated. Similar procedures were used for a propane tank.

In order to calculate the radiation interchange between the propellants, the tanks were positioned as shown in the configuration. Using the same programs as above, radiation configuration factors between a propane sphere and a flox sphere were determined and a radiation heat balance was established.

Calculations were made to determine the conduction heat transfer from the warm payload down to the aluminum struts, through the fiberglass tank supports into the propellants. In making these calculations, the thermal conductivities corresponding to the mean temperatures of the struts and tank supports were used and no attempt was made to account for thermal contact resistance at any structural joint.

From the above analyses, the results show that the net heat transfer to the propane was negative and venting was unnecessary except when the propane storage temperature was below 250°R. However, in the case of the flox, even though the radiation heat transfer could be overcome by installing more than 1.5 inches of insulation, the conduction heat transfer made the possibility of non-venting very marginal.

Two Oblate Spheroid Configuration

In calculating the net radiation heat transfer, it was assumed that the tanks were positioned as shown and radiation interchange occurred between the payload disc and the propane, between the propane and flox, and from the bottom half of the flox tank to a 0°R environment. The calculation procedures used were the same as in the four sphere configurations.

For this configuration, as with the four sphere case, temperature averaged thermal properties were used to estimate the conduction heat transfer rates into the propellants.

The results of the above analyses indicate that with a one-half inch thickness of foam insulation on both tanks (primarily for ground hold considerations), this configuration can be maintained in a non-vented condition so long as the stage is oriented payload to the sun and is not subjected to other thermal sources.

The resulting weight penalty of the above insulation is shown in figure 7 as a function of propellant weight.

Propulsion

Engines

The assumed variations of engine weight with engine thrust are shown in figure 8. Chamber pressure and nozzle expansion ratio were not varied in the study since previous unpublished in-house studies have shown the values indicated on figure 8 to be about optimum.

In arriving at the engine weights, pure regenerative cooling was assumed for the pump-fed engines and a combination of regenerative plus transpiration cooling for the pressure-fed engines.

The weight estimations are for the complete engine package including the weights of the engine gimbal and actuator systems.

Delivered specific impulse values of 390 seconds and 365 seconds were used for the pump-fed and pressure-fed engines, respectively. These values are based on the use of an oxidizer composition of 70 percent fluorine and 30 percent oxygen and an engine mixture ratio of 4.5 to 1.

Pressurization

The use of a cold gas pressurization system was assumed for the pressurization and expulsion of all propellant tanks. The helium initial storage conditions were 520°R and 4000 psia. The final helium bottle pressure was 100 psi above the tank operating pressure. In calculating the pressurant requirements, the effects of heat transfer between the bottle and the helium in it, as well as the heat transfer between the propellant tanks and the incoming pressurant, were included.

Propellant Utilization

It is highly unlikely that an active propellant utilization system would be used on a stage of this size. The small reduction in jettison weight would not justify the increased complexity. Rather, a calibrated propellant system with a fuel bias to minimize propellant outage seems more realistic. In this study, detailed calculations involving the many parameters that affect propellant outage were not made, but the propellant system weight shown on the weight tabulations includes a 50-pound weight penalty to account for the propellant outage associated with a calibrated propellant system.

Launch Vehicle

The previous high energy upper stage studies reported in reference 1 indicated that for the two missions of interest here (a high energy solar probe and a low energy planet orbiter) a launch vehicle of the Saturn IB-Centaur class was desirable. Therefore, all upper stage mission performance data generated in this study were based on the use of the Saturn IB-Centaur. The results and comparisons would be similar if the Titan IIIC-Centaur were used. In all cases, the Saturn IB guidance system was used to guide the first two stages of the vehicle. For the solar probe mission, it was assumed that the Centaur guidance system was removed and that guidance for both Centaur and upper stage operation was provided by the upper stage guidance system. For the long coast planetary orbiter mission, it was assumed that upper stage guidance would be provided by the spacecraft (i.e., payload). Centaur guidance was retained to provide guidance during Centaur operation. Flight performance reserve was provided in both the S-IVB stage and the Centaur stage to account for performance dispersions. A 100 n.mi. parking orbit was assumed in all cases with a coast time of up to 30 minutes allowed between the two Centaur burn periods. For mission performance calculations, a weight of 8900 pounds was assumed for the 260-inch shroud that would enclose the payload, upper stage and Centaur stage during boost. In addition, a weight of 390 pounds was assumed for the interstage between Centaur and the upper stage and a weight of 740 pounds for that between Centaur and the S-IVB.

STAGE SIZING AND PERFORMANCE

Utilizing the jettison weight and engine weight curves discussed previously, calculations were made to determine the effect of stage propellant loading and thrust level on payload capability. The results for the solar probe mission are presented in figure 9. For the pump-fed stage, figure 9a, a stage with 8000 to 9000 pounds of propellant and an engine thrust level of 10,000 pounds is near optimum. For the pressure-fed stage, figure 9b, a propellant weight of 4000 to 5000 pounds with a thrust level of 5000 pounds appears attractive.

The solar probe payload capability of stages utilizing propane-flox is compared, in figure 10, to that obtainable using other propellant combinations. The performance data presented here and in subsequent figures for the earth storable stage and the hydrogen-fueled stages are based on the work reported in reference 1 with some modifications of the stage weights to reflect recent design effort and to make the performance ground rules consistent with those for the propane-flox stages. As would be expected on a short coast mission, the performance with propane-flox is better than that of earth storables and poorer than that of the deep cryogenics. It is interesting to note, however, the substantial performance advantage of the pump-fed propane-flox stage over that of the pressure-fed stage. The pump-fed performance advantage is due to a combination of lower stage jettison weight and higher engine specific impulse. At a propellant weight of 3500 pounds, about 70 percent of the advantage is due to the higher engine specific impulse (390 pump-fed versus 365 pressure-fed), while at a propellant weight of 9000 pounds, only 40 percent of the payload difference is due to the higher specific impulse. At the present time, insufficient experimental data exists to precisely predict the specific impulse efficiencies to expect from propane-flox engines. From the above comments, however, it is evident that even with substantial variation in the assumed specific impulse levels, the pump-fed stage would outperform the pressure-fed stage.

The effects of both stage size and thrust level for a low energy, long storage time mission, such as a Mars orbiter, are smaller than for the solar probe mission. For example, the Voyager design studies indicate an optimum vehicle thrust/weight ratio of less than 0.20. However, if one assumes (as was done for the cryogenic stages in reference 1) that a propane-flox stage would serve as a multipurpose stage, then a thrust level more consistent with the solar probe requirements would appear desirable. Therefore, the payload data for a planet orbiter mission as presented in figure 11 and figure 12 for pump-fed propane-flox stages are based on an engine thrust level of 10,000 pounds. Also, as was pointed out earlier, for the long coast mission, the upper stage guidance system was removed and guidance provided by the payload. However, due to the longer coast time involved, additional attitude control propellant is required. In addition, as with the deep cryogenic stages of reference 1, a monopropellant midcourse correction propulsion system was added to the stage. This permits the primary propulsion system to remain sealed during the long coast to Mars. A midcourse correction capability of 150 feet per second was provided utilizing a hydrazine monopropellant system with a specific impulse of 240 seconds. The net effect of these changes was that the stage jettison weight increased 248 pounds in going from the solar probe mission to the planetary orbiter mission.

The effect of stage size on payload capability for a Mars orbiter mission is presented in figure 11 for both a pump-fed propane-flox stage and a pump-fed hydrogen-oxygen stage. Payload capability is shown versus stage propellant capacity for two Mars orbits, circular, figure 11a, and highly elliptic, figure 11b, and for two payload situations, all payload into orbit or half into orbit and half into a landing capsule that is ejected prior to retro into the Martian orbit. The symbols indicate the discrete propellant weights required to accomplish the mission based on an earth injection weight of 10,500 pounds. The curves present performance for stages with a larger propellant capacity, but off-loaded to the required propellant weight. The insensitivity of payload to stage size is evident from the flatness of the curves.

The hydrogen-oxygen data were generated assuming hydrogen venting could occur whenever required with no weight penalty associated with settling propellants prior to each vent. If a substantial weight penalty were required in order to insure an adequate venting capability, then the hydrogen-oxygen payload levels shown in figure 11 would be reduced accordingly. On the other hand, the hydrogen-oxygen stage was constrained to use the RL-10 engine. A similar constraint on the propane-flox stage would reduce the payload levels shown in figure 11 for propane-flox by about 175 pounds, substantially reducing the performance margins shown in figure 11.

For this same Mars orbiter mission, a comparison of the payload capability obtainable with a pump-fed propane-flox stage to that obtainable using either a pump-fed hydrogen-oxygen stage or a pressure-fed, payload integrated storable stage is presented in figure 12.

Payload is plotted versus apofocus altitude for a constant perifocus altitude of 1000 n.mi. The curves for the storable stage were generated by resizing the stage to carry just the right amount of propellant to accomplish the mission. The curves for the hydrogen-oxygen and propane-flox stages were generated assuming stage sizes of 7000 and 9000 pounds, respectively, off-loaded to the proper propellant weights.

The propane-flox stage exhibits a 5 to 8 percent greater payload capability than the hydrogen-oxygen stage depending upon the apofocus altitude and percent payload in orbit. For highly elliptic orbits (20,000 n.mi. apofocus), the propane-flox margin over the earth storable is slight, 0 to 5 percent. For the more energetic, circular orbit case (1000 n.mi. apofocus), this margin increases to 8 to 16 percent. These modest performance margins over earth storables are the result of the low ΔV requirements for retro into a Mars orbit. In figure 12, the ΔV for the retro maneuver varied from 4500 feet per second to 7600 feet per second.

CONCLUDING REMARKS

An analysis was conducted to determine the space storability and payload capability of small propane-flox upper stages for use on NASA unmanned missions. Both pump-fed and pressure-fed stages were investigated. Two mission types were considered: (1) a high energy, short coast mission (solar probe), and (2) a low energy, long coast mission (planetary orbiter).

Considering the short coast, high energy mission first, the results indicate that pressure-fed propane-flox stages are not attractive for this type mission. The pump-fed stages far outperform them. In addition, a pump-fed propane-flox stage incorporating a new engine properly sized for the high energy mission does not perform as well as cryogenic stages constrained to use the RL-10 engine. The propane-flox performance approaches that of the hydrogen-oxygen stage, but falls considerably short of that of the hydrogen-fluorine stage. Going to new engines on the cryogenic stages or constraining the propane-flox stage to use an RL-10 derivative would increase the performance gap.

Considering the long coast mission, the results indicate it is possible to design a propane-flox stage having a long space storage (no vent) capability. To accomplish this requires proper selection of stage configuration, vehicle orientation, propellant storage temperatures and insulation (multilayer and foam only; shadow shields or refrigeration devices were not considered in the study). The resulting performance potential, however, does not appear to be significantly better than that obtainable with presently existing storable propellants; at least not for a low energy Mars orbiter type mission. If zero g venting of cryogenics proves difficult or has large weight penalties associated with it, then for high energy, long storage missions a propane-flox stage may be desirable.

While this analysis was concerned with the space storability of the propane-flox propellant combination, the results should be applicable, essentially unchanged, for the butene 1-flox combination. For methane-flox, however, the methane would have to be stored at about 250°R in order for the radiation heat transfer from the tank to equal the total heat transfer into the tank. The corresponding vapor pressure of the liquid at this temperature is about 75 psia and this, plus the decreased liquid density, would impose a stage penalty in the form of increased tankage and pressurization system weight.

REFERENCES

1. NASA TMX-52127, An Analysis of Chemical Upper Stages for NASA Scientific Missions, AD&E Division, Lewis Research Center, 1965.

2. Linde Company Super Insulation Applied to Space Vehicles, Lindquist, C. R., December 1, 1962.

TABLE I

Weight Summary for a 6000-Pound Propane-Flox
Kick Stage, Pump-Fed Engine
Four Sphere Configuration

Pressurization		64
He Tank	39	
Helium	5	
Hardware	20	
Propulsion		460
Engine	175	
Fuel Tank	35	
Oxidant Tank	87	
Insulation	53	
Prop. System	110	
Structure		250
Guidance and Autopilot		150
Electrical		105
Attitude Control		50
Separation		22
Telemetry		85
Range Safety		15
Residuals		60
Contingency		138
Burnout Weight		1399
Impulse Propellant		6000
Stage Total Weight		7400
Stage Mass Fraction		0.810

TABLE II

Structural Weight Summary for a 6000-Pound Propane-Flox
Kick Stage, Pump-Fed Engine
Four Sphere Configuration

Structure		250
Payload Support Structure	25	
Cruciform Beam	87	
Cruciform Beam Ring	54	
Propellant Tank Supports	44	
Propellant Tank Brackets	14	
Helium Bottle Support	7	
Engine Truss	19	

TABLE III

Weight Summary for a 6000-Pound Propane-Flox
Kick Stage, Pump-Fed Engine
Two Oblate Spheroid Configuration

Pressurization		64
He Tank	39	
Helium	5	
Hardware	20	
Propulsion		406
Engine	175	
Fuel Tank	31	
Oxidant Tank	71	
Insulation	19	
Prop. System	110	
Structure		229
Guidance and Autopilot		150
Electrical		105
Attitude Control		50
Separation		22
Telemetry		85
Range Safety		15
Residuals		60
Contingency		131
Burnout Weight		1317
Impulse Propellant		6000
Stage Total Weight		7317
Stage Mass Fraction		0.820

TABLE IV

Weight Summary for a 6000-Pound Propane-Flox
Kick Stage, Pump-Fed Engine
Two Oblate Spheroid Configuration

Structure		229
Forward Ring	28	
Aft Ring	33	
Struts	56	
Propellant Tanks Support	26	
Engine Thrust Cone	74	
Helium Tank Support	12	

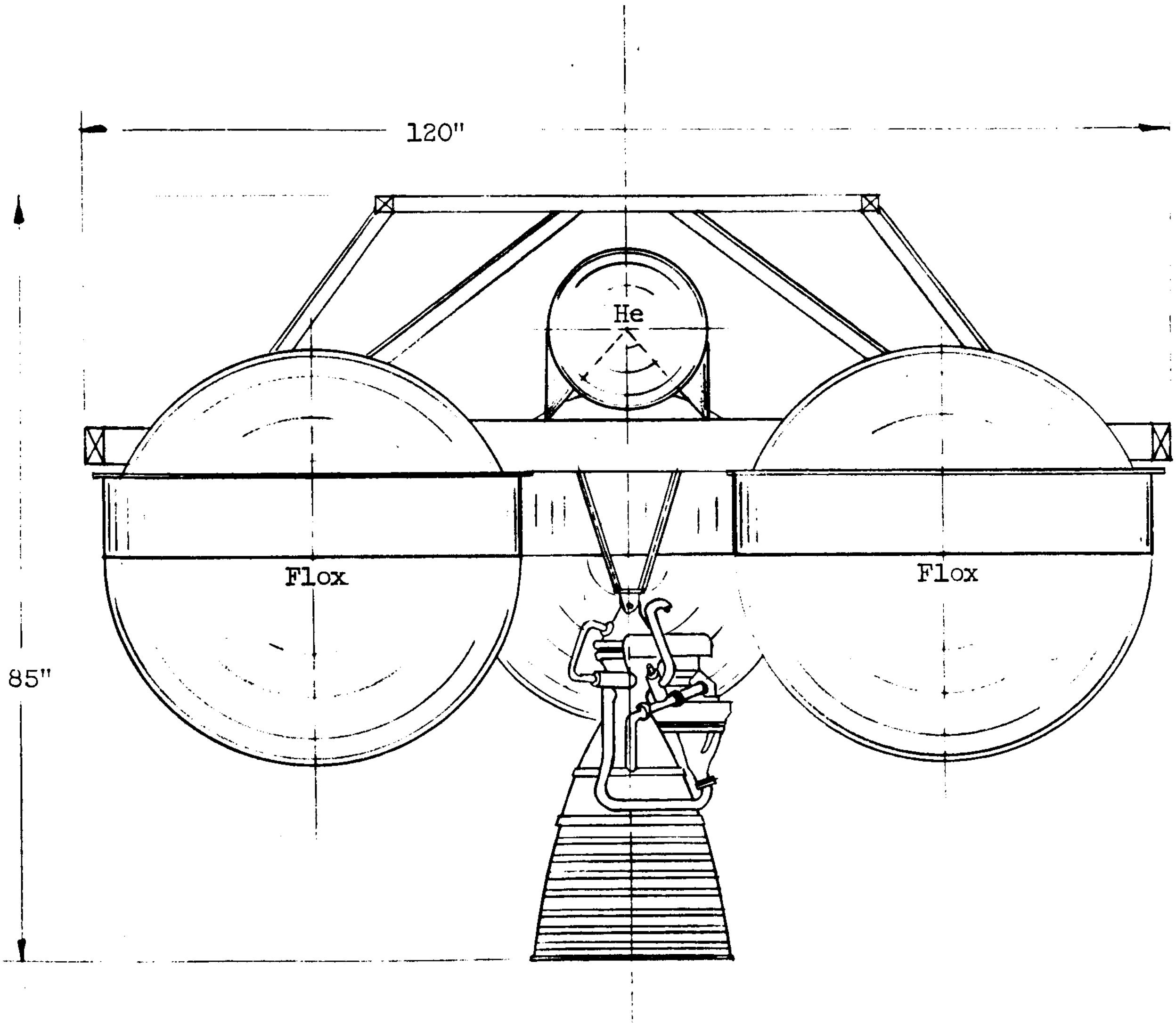


Figure 1. - Pump fed propane-flox stage, four sphere configuration.

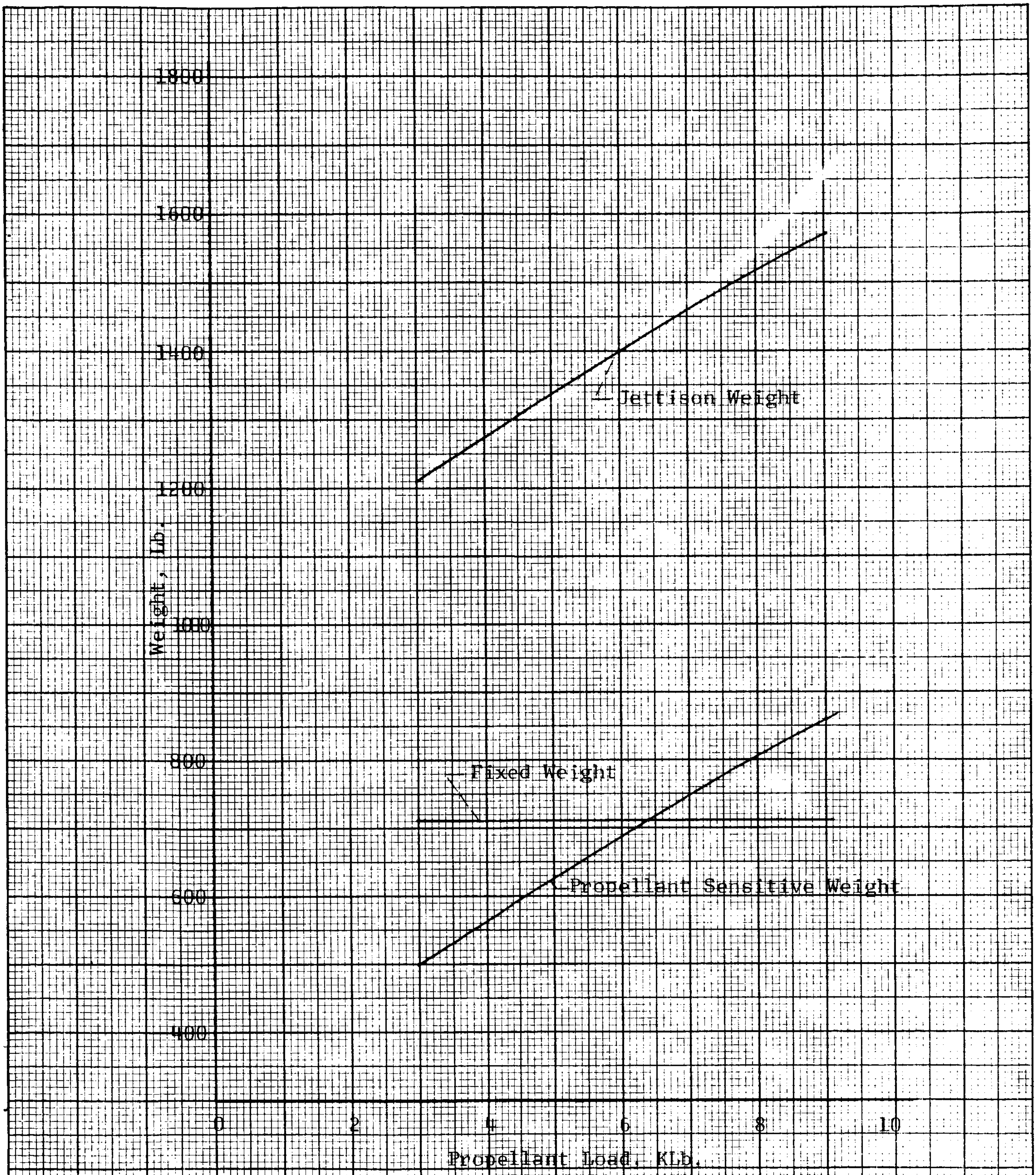


FIGURE 2 - Jettison Weight Breakdown for Propane-Flox Kick Stage, Four Spheres Configuration; Pump Fed.

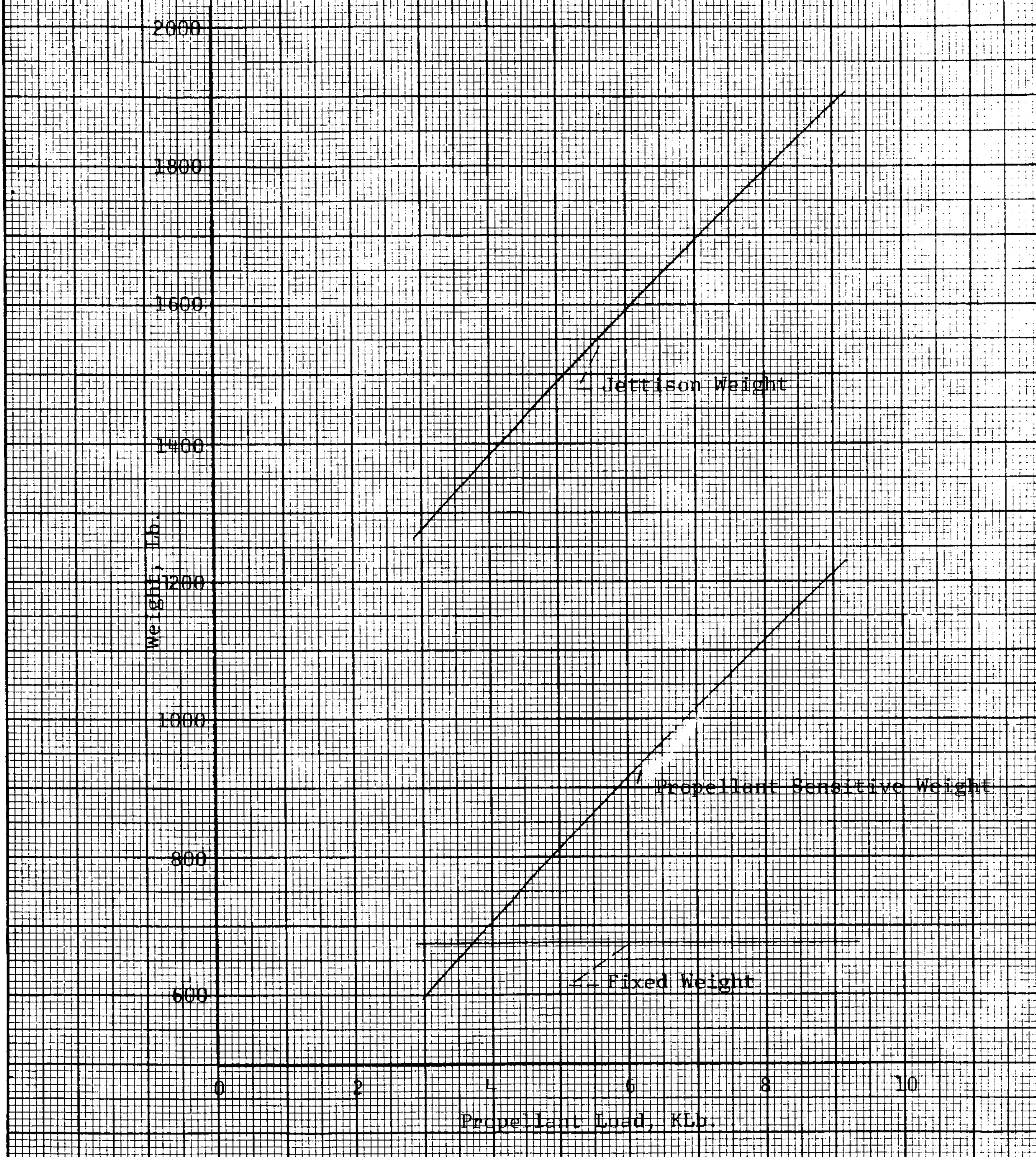


FIGURE 3 - Jettison Weight Breakdown for Propane-Flox Kick Stage. Four Spheres Configuration; Pressure Fed.

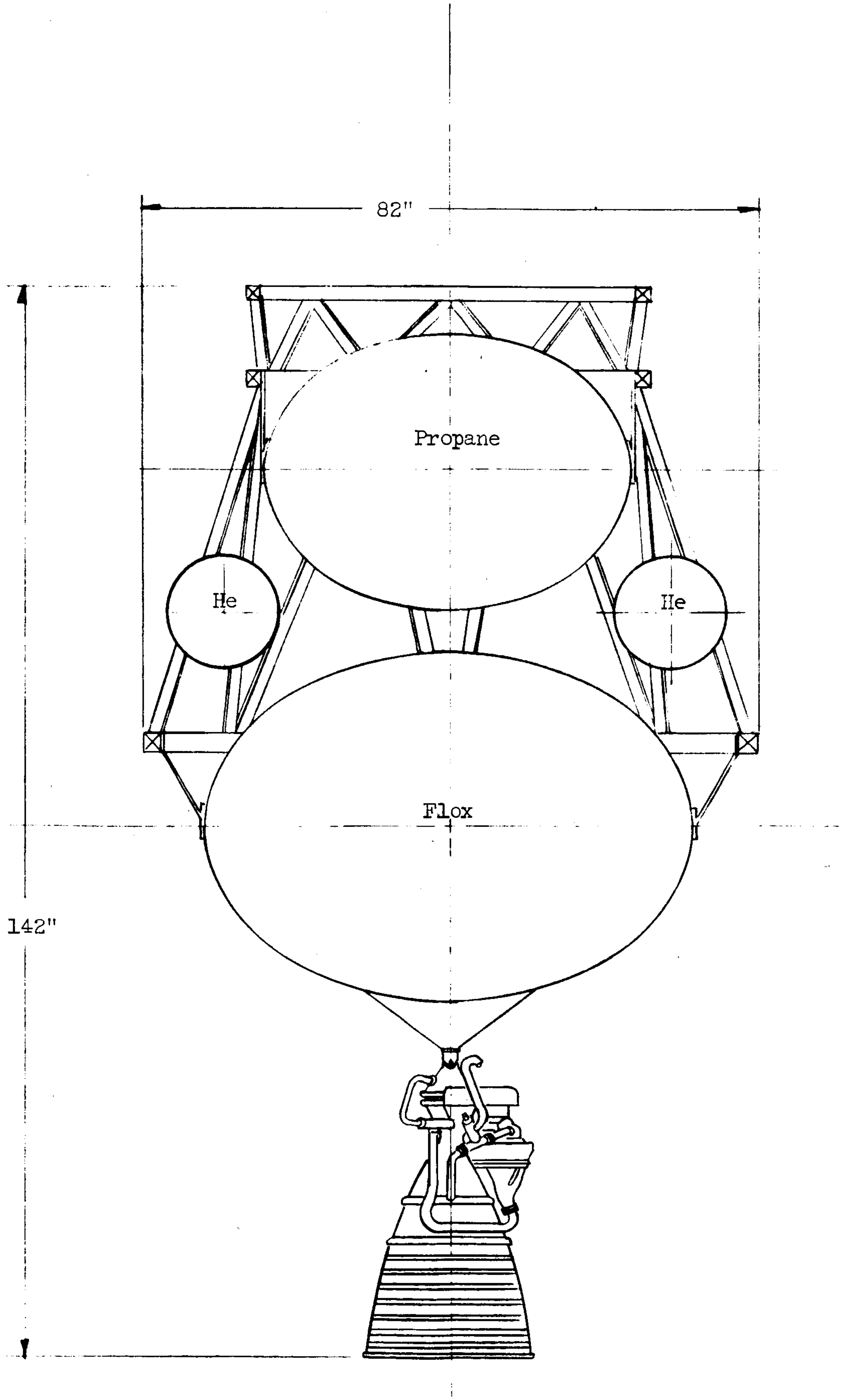


Figure 4. - Pump-fed propane-flox stage, two oblate spheroid configuration.

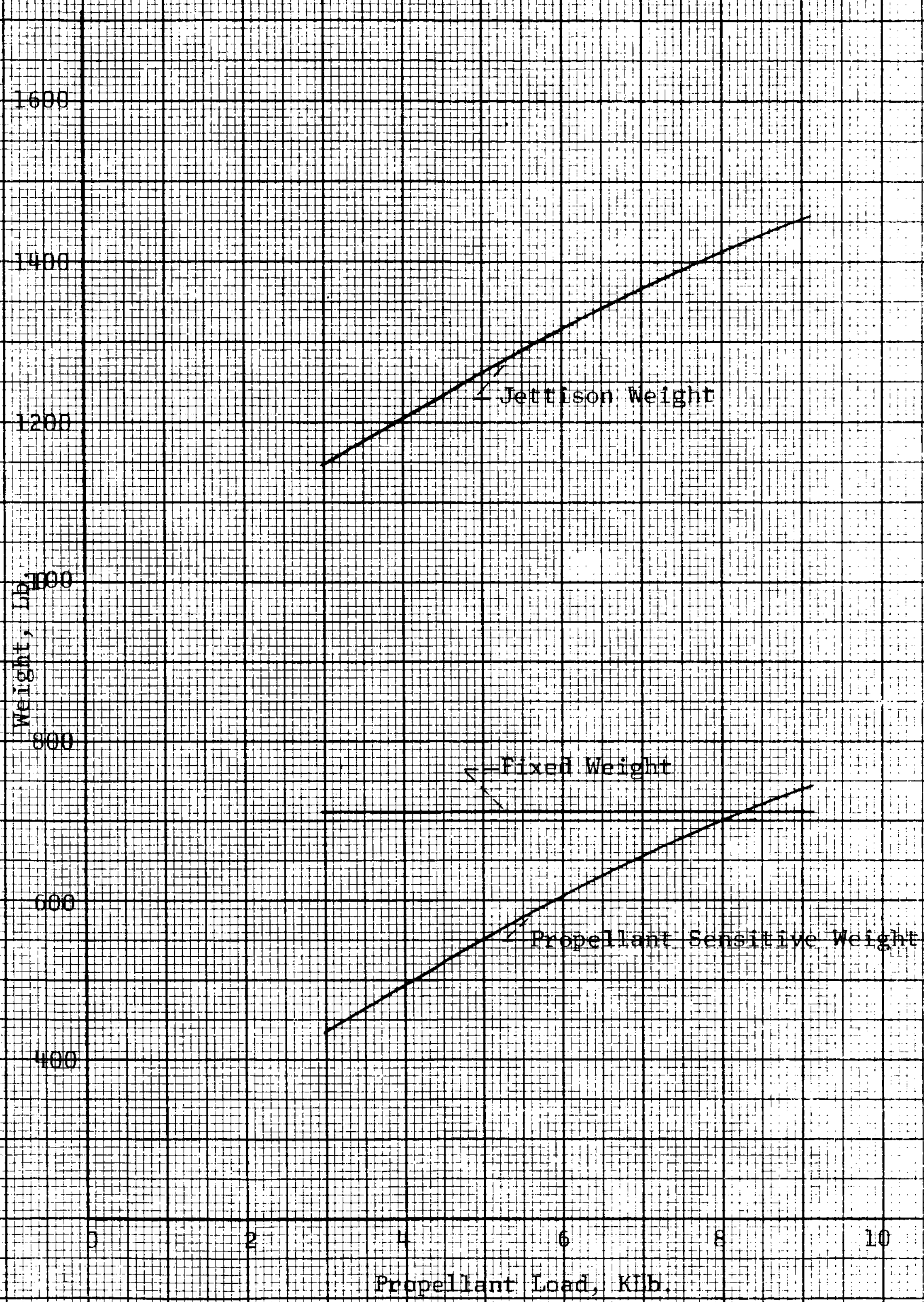


FIGURE 5 - Jettison Weight Breakdown for Propane-Flox Kick Stage. Two Oblate Spheroids Configuration; Pump Fed.

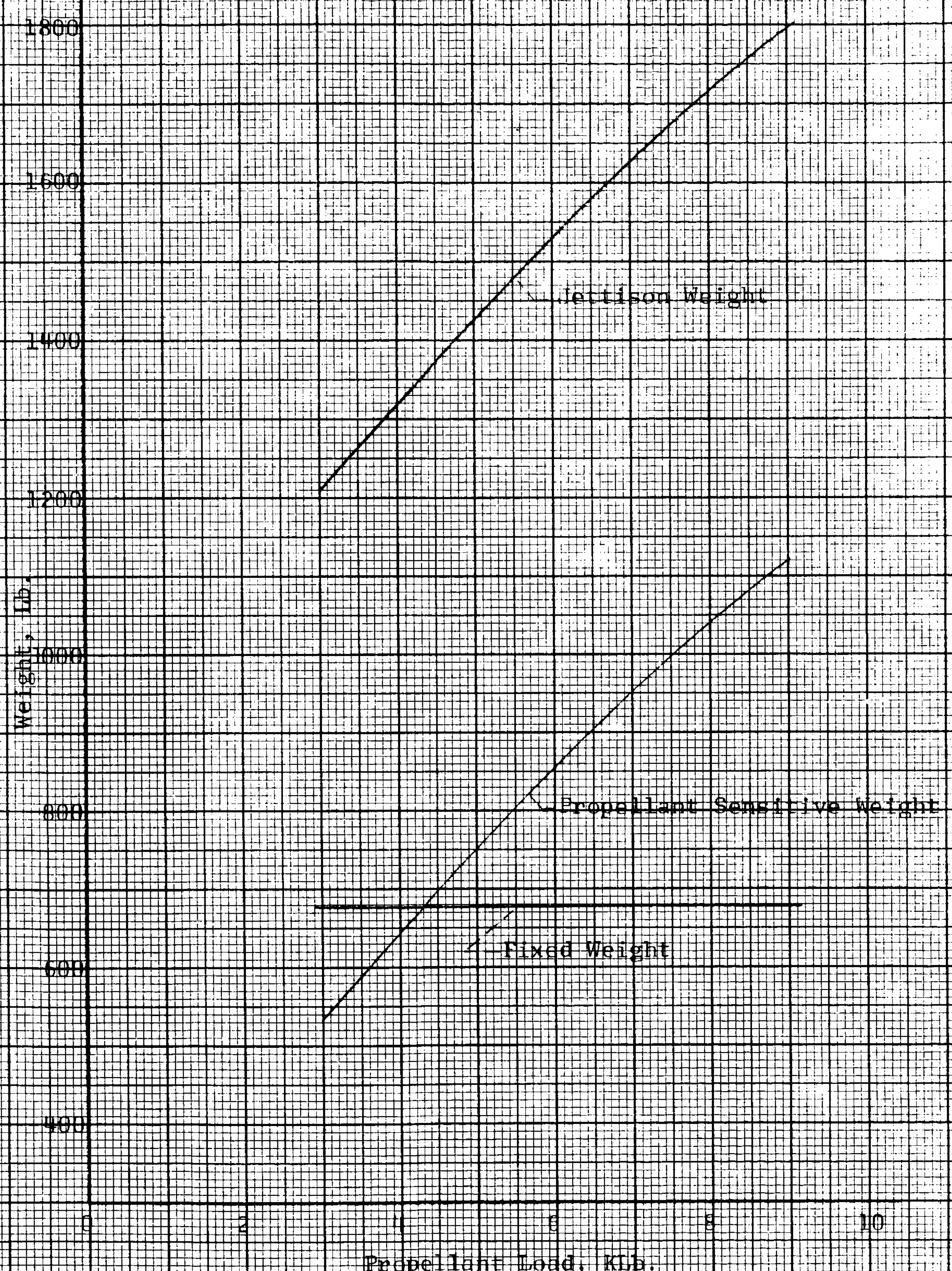


FIGURE 6 - Jettison Weight Breakdown for Propane-Flox Kick Stage. Two Oblate Spheroids Configuration; Pressure Fed.

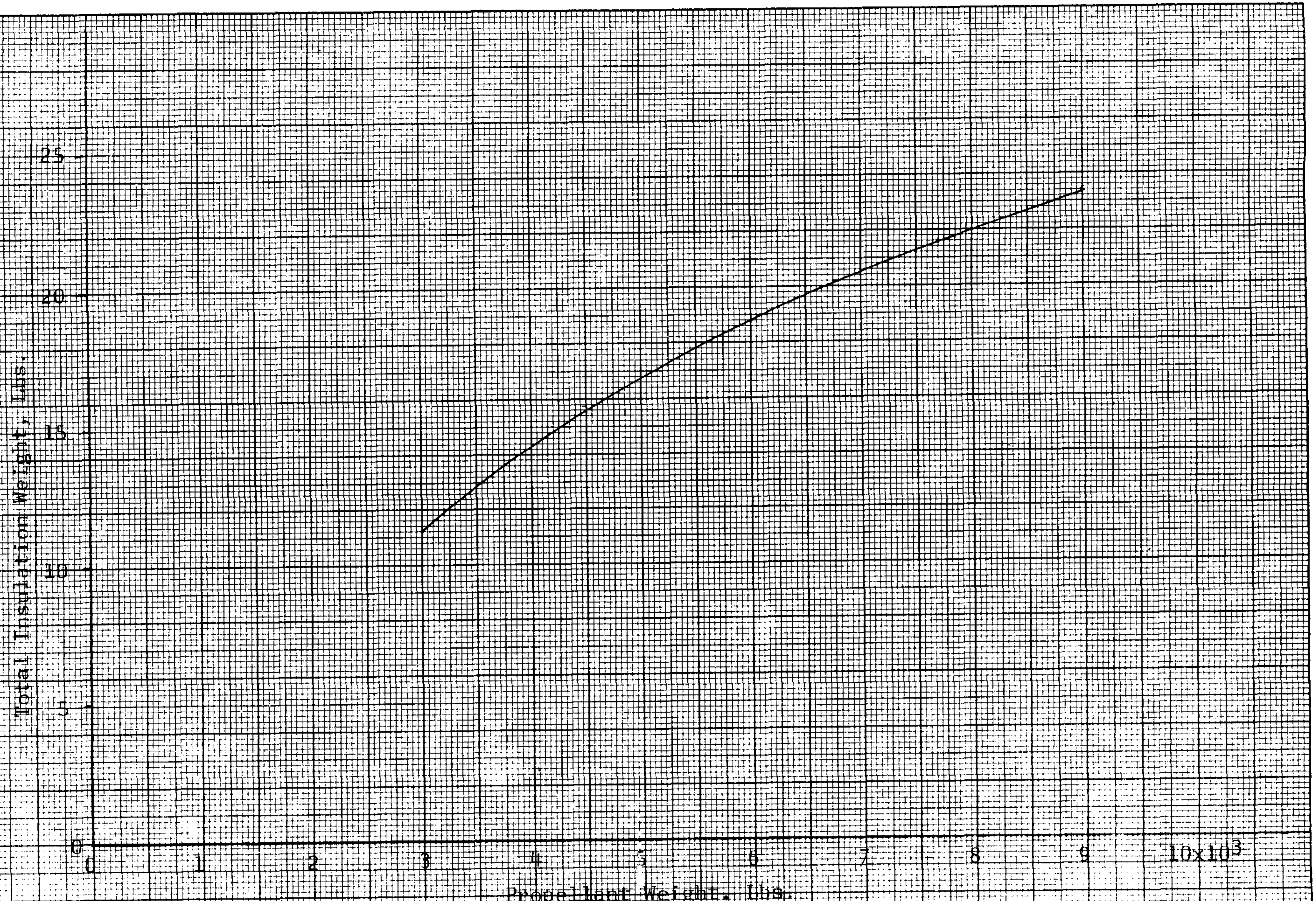


FIGURE 7 - Insulation Weight Vs. Propellant Weight for Propane-Flox. No Boiloff.
Two Oblate Spheroid Configuration, Sun Oriented.

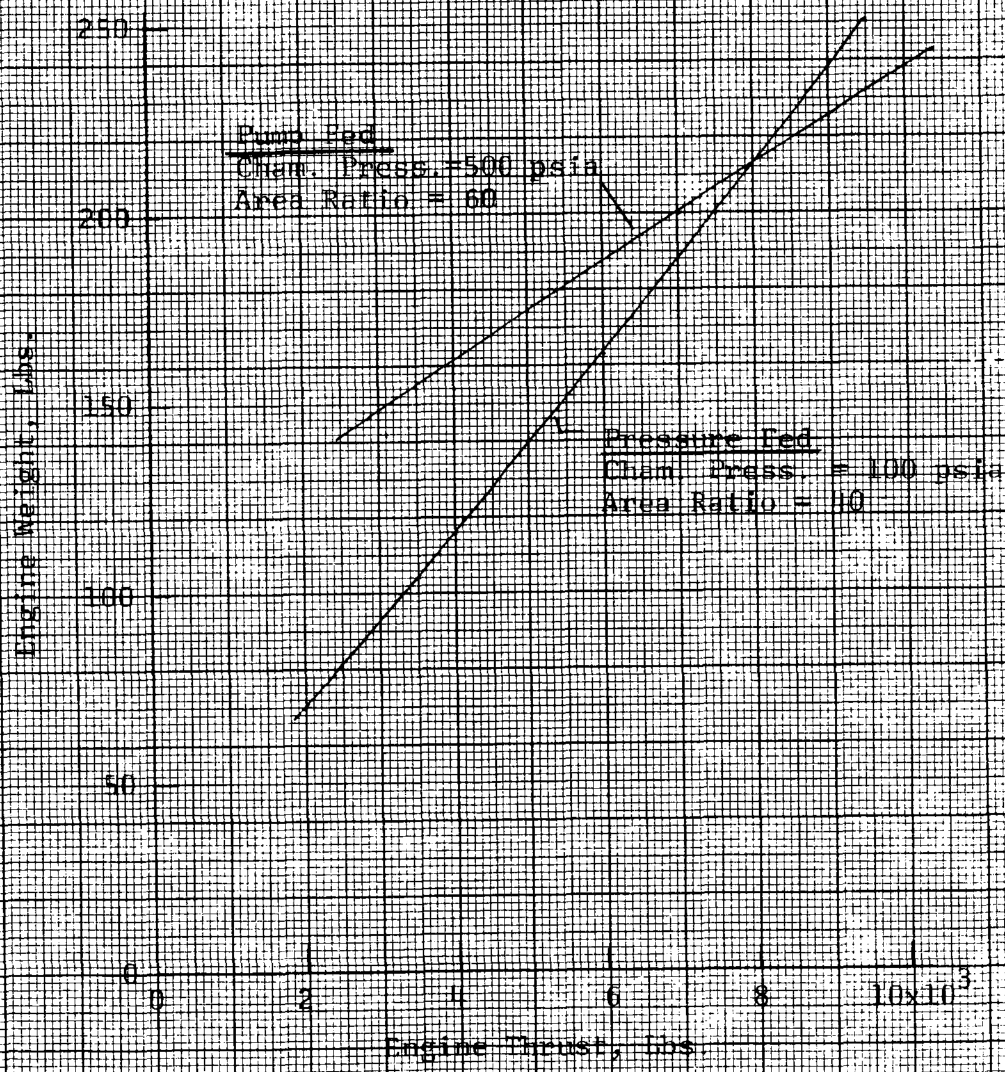


FIGURE 8 - Assumed Engine Weights Vs. Thrust Level for Propane-Flox Engines

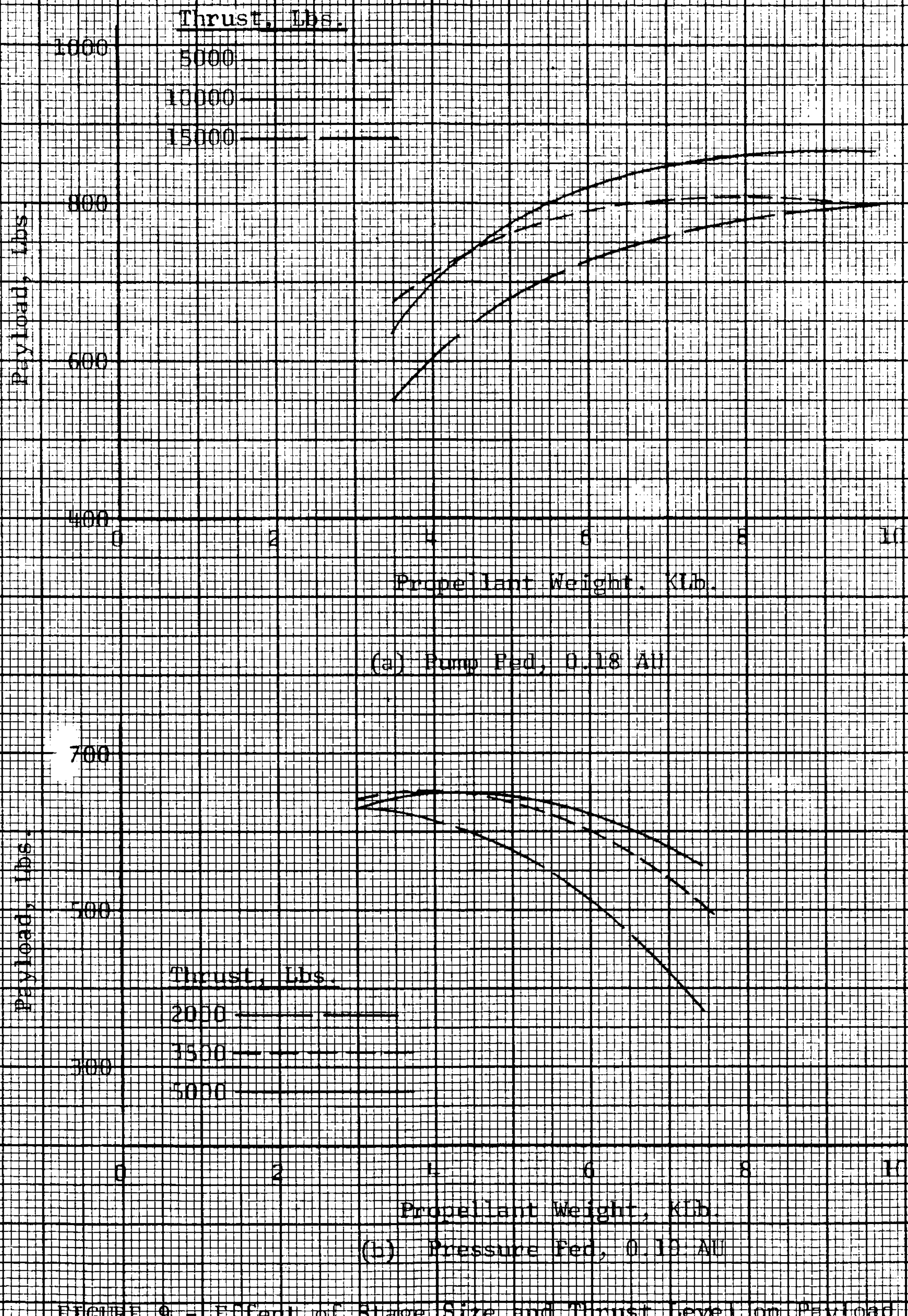


FIGURE 9 - Effect of Stage Size and Thrust Level on Payload Capability. Propane-Flox; Solar Probe Mission; Saturn IB-Centaur Boost

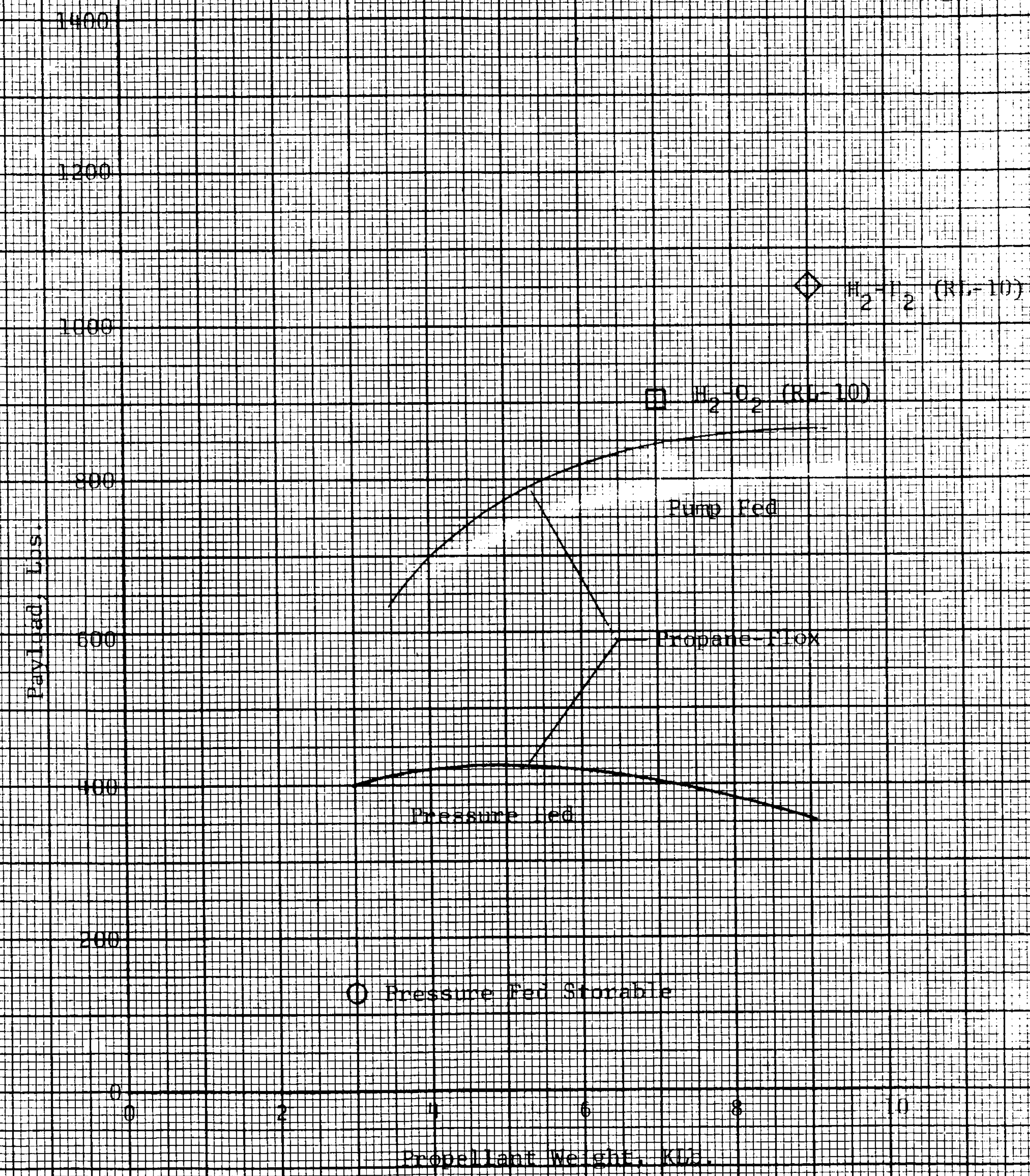
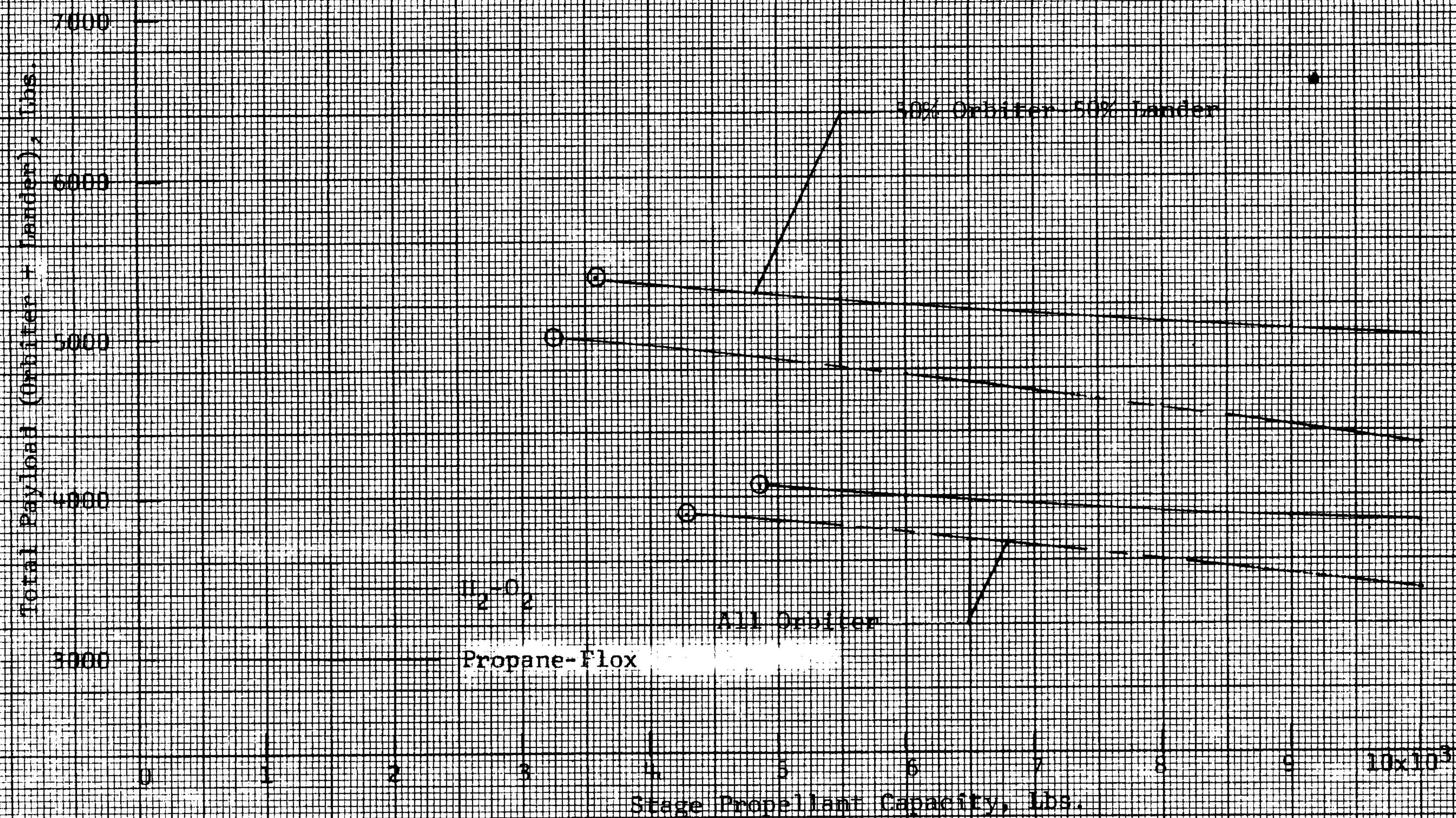
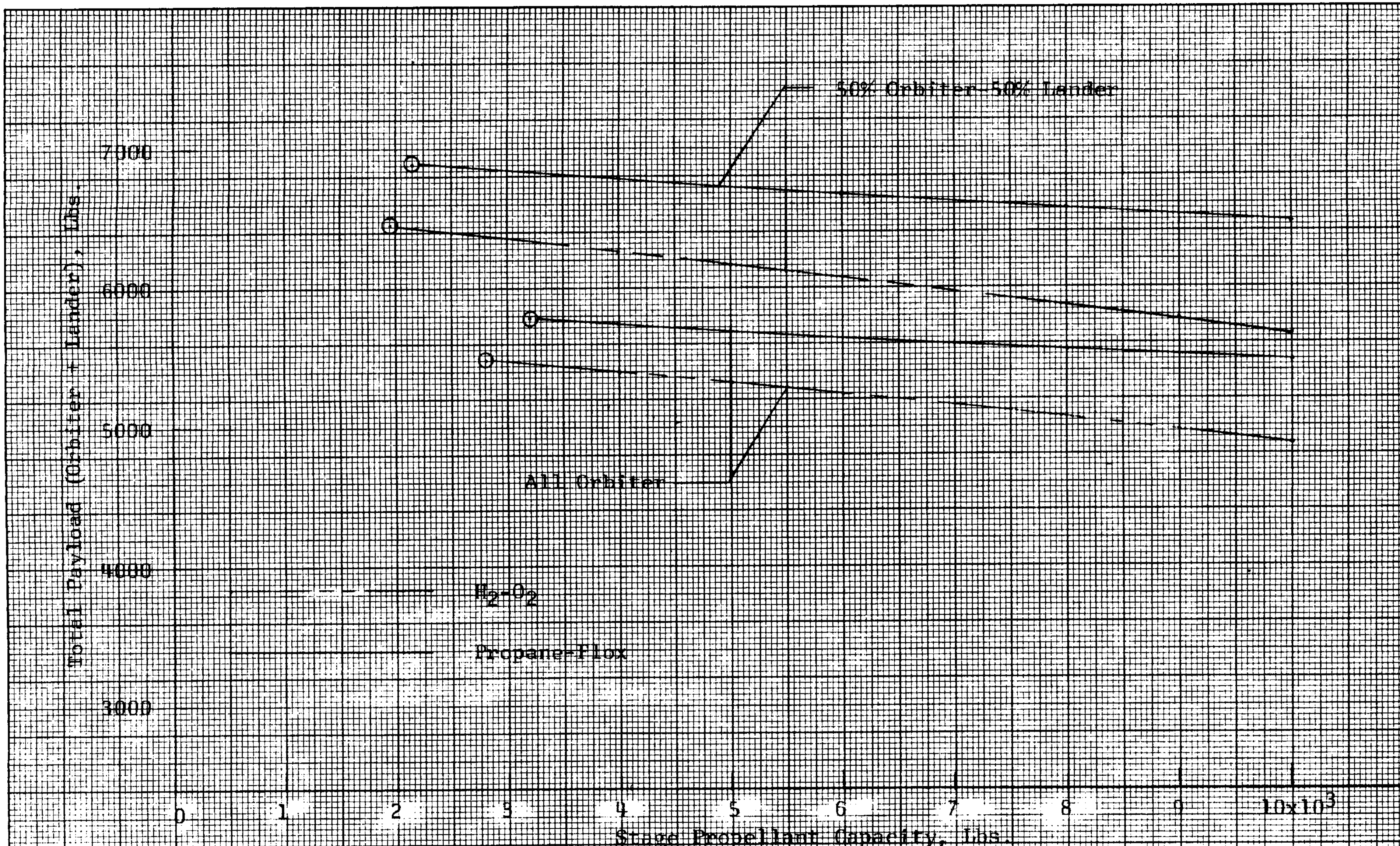


FIGURE 10 - Comparison of Propane Flox Upper Stage Performance with Other Upper Stages. 0.18 A.U. Solar Probe; Saturn IB-Centaur Launch



a) 1000 n.mi. Circular Orbit

FIGURE 11 - Effect of Propellant Capacity on Payload Capability of H_2-O_2 and Propane-Flox Stages for Mars Orbiter Mission. Pump-fed Stages, 1971, 60-day Opportunity, Saturn IB-Centaur Boost.



b) 1000 x 20,000 n.mi. Elliptical Orbit

FIGURE 11 - Effect of Propellant Capacity on Payload Capability of H_2-O_2 and Propane-Flox Stages for Mars Orbiter Mission. Pump-fed Stages, 1971, 60-day Opportunity, Saturn IB-Centaur Boost

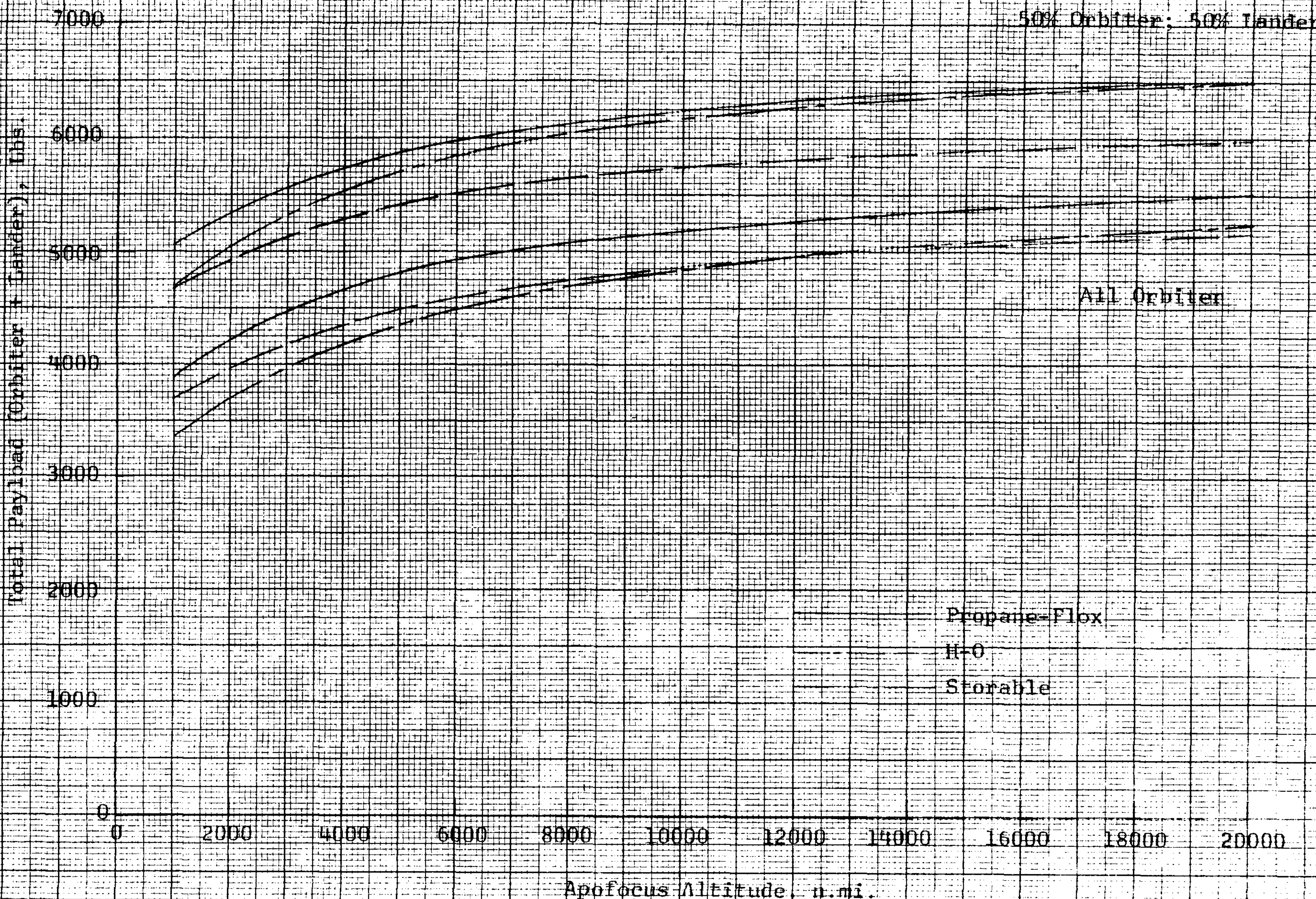


FIGURE 12 - Comparison of Payload Capability of Propane-Flox, Hydrogen-Oxygen, and Earth Storable Stages for Mars Orbiter Mission. 1971, 60-Day Opportunity, Perifocus Altitude - 1000 n.mi., Saturn IB-Centaur Boost